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GUIDANCE AND NAVIGATION REQUIREMENTS FOR UNMANNED FLYBY AND SWINGBY MISSIONS TO THE OUTER PLANETS

October 1971

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GUIDANCE AND NAVIGATION REQUIREMENTS FOR UNMANNED FLYBY AND SWINGBY MISSIONS TO THE OUTER PLANETS

(Summary of Contract NAS-2-5043)

VOLUME I Summary Report

OCTOBER 1971

CHARLES STARK DRAPER LABORATORY

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

CAMBRIDGE, MASSACHUSETTS

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FOREWORD

This report summarizes the results of studies conducted under contract NAS2-5043, on the subject of Navigation Requirements for Unmanned Flyby and Swingby Missions to the Outer Planets. The studies were conducted by the Charles Stark Draper Laboratory of the Massachusetts Institute of Technology. Study results are organized into the following four volumes:

Volume-I. Summary Report,
Volume-II. High Thrust Missions,
Volume-III. Low Thrust Missions.
Volume-IV. High Thrust Missions - Part 2

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INTRODUCTION

Unmanned spacecraft missions to the outer planets are of current interest to planetary scientists, and are being studied by NASA for the post 1970 time period. Flyby, entry and orbiter missions are all being considered using both direct and planetary swingby trajectory modes. The study summarized in this volume is concerned with navigation and guidance requirements for a variety of missions to the outer planets and comets including both the three and four planet Grand Tours.

Funded under NASA contract NAS2-5043, and directed by the Advanced Concepts and Missions Division of the Office of Advanced Research and Technology, this study was divided into three phases. Phase A, reported in Volume II, studies the requirements for guidance and navigation on missions in which the midcourse trajectory corrections are made exclusively by means of short-duration, impulsive, velocity changes, termed "high thrust" in this report. Phase B, reported in Volume III, studies the same requirements problems for planetary orbiter missions where the spacecraft uses continuous, low-thrust propulsion for most of the mission. The orbit insertion maneuver, assumed to be impulsive, was not considered part of this study. Phase C, reported in Volume IV extends the techniques developed under Phase A to Comet Rendezvous, Jupiter Entry, and Three-Planet Flyby missions. Overall study objectives, as listed by NASA's OART Advanced Concepts and Missions Division were:

(1) Determine the characteristics associated with (a) totally onboard, (b) totally Earth-based, and (c) a combination of Earth-based and onboard navigation concepts.

(2) Determine the associated navigation and guidance subsystems weight, volume, and power estimates for representative navigation and guidance subsystem concepts applied to mission objectives.

(3) Determine the accuracy requirements placed upon the midcourse propulsion and attitude control subsystems by each of the combinations above.

(4) Perform tradeoff analyses which compare, on a total guidance and navigation subsystem basis, the three navigation concepts for each nominal mission, considering both the interplanetary and near-planet portions of the mission.

Specific missions analyzed were:

Phase A — High Thrust

1) 1973 Jupiter Flyby
2) 1977 Jupiter Swingby to Saturn
3) 1977 Grand Tour (Jupiter, Saturn, Uranus, Neptune)

Phase B — Low Thrust

1) 1979 Jupiter Orbiter
2) 1980 Saturn Orbiter

Phase C — High Thrust

1) 1985 Rendezvous with Comet P/Tuttle-Giacobini-Kresak
2) 1989 Rendezvous with Comet P/Tempel 2
3) 1978 Jupiter 800 Day Entry Mission
4) 1978 Jupiter 1200 Day Entry Mission
5) 1979 Jupiter-Uranus-Neptune Flyby
6) 1977 Jupiter-Saturn-Pluto Flyby

To fulfill the listed objectives, two statistical computer simulations utilized patched-conic nominal trajectories, and impulsive velocity corrections. Error analyses for phases A and C were performed about the nominal reference trajectories by linearizing the equations of motion and using only the statistics of first-order deviations from the reference solution, then applying the techniques of linear filter theory. The Phase B simulation was similar to this except the additional terms resulting from the low-level constant thrust required numerical integration of the equations of motion. In addition to these major simulations, a number of other computer programs were generated for various purposes including guidance scheme analysis and trajectory production.

Using these simulations, parametric studies of spacecraft position, velocity, and planetary ephemeris errors were generated with Deep Space Network and onboard navigation system errors as parameters. From these parametric results conclusions are drawn about the guidance and navigation requirements for outer planet missions, and about the effects these requirements have on onboard sensor design.

MODELING AND SIMULATION STUDIES

STATISTICAL SIMULATION—HIGH-THRUST CASES

The entire navigation and guidance requirements study was performed by considering only the statistics of first-order deviations from a reference trajectory; thus all the techniques of modern linear filter theory could be employed in the analysis. By making the approximation that all random processes are Gauss-Markov processes, only second-order statistics were necessary and it was possible to obtain recursion formulas for the filter which were extremely convenient for use on a digital computer. The reference trajectory used throughout was the nominal mission trajectory; measurements were linearized about nominal values which were computed using the reference trajectory.

Biases occur in the simulation in several places. The masses of the outer planets are imperfectly known and thus introduce a bias into the trajectory dynamics.* The Deep Space Network is modeled with bias uncertainties in longitude and distance from the earth's spin axis, and the accelerometers have an associated null bias error.

* the planet radii are considered to have a bias uncertainty (which is filtered). The onboard navigation sensor is modeled with white noise and a measurement bias.

The artifice commonly used to avoid biases in a linear filtering problem is to adjoin the biases to the state and estimate them. However, if the estimation error does not have a lower bound, the biases are soon known perfectly in contrast to many real situations where there is a bias drift that precludes perfect bias knowledge. Therefore it was necessary to develop a filter theory for considering these biases in an optimal manner for the problem of interest. This was done by identifying the necessary cross correlation terms, and developing recursion formulae which were used to adjust the biases at each step of the simulation.

Central to the error analysis computation procedure is the fact that the mission under consideration is divided into a number of decision points. The frequency, spacing, and total number of these points is completely flexible and is specified at run time. At each point, a decision is made by the computer whether or not to make a velocity correction, whether or not to process a DSN measurement, and what onboard measurement or measurements to take, if any.

After the measurement decision process has been completed for a given time and position in the mission, and the new covariance matrices have been computed, the process is repeated at the next measurement decision point with covariance matrices extrapolated from the previous point. At selected points in the mission (generally chosen to be the planetary sphere of influence and periplanet) the error covariances are read out and used for the systems studies.

MODELING—HIGH-THRUST MISSIONS

Since the simulation is concerned mainly with navigation measurements, the models of primary importance are those of the ground-based Deep Space Network and the spacecraft-based navigation sensor. Because of the location of deep space radio tracking stations on a rotating earth, the complete dynamics of spacecraft motion can be obtained from a single range rate equation. This equation contains the geometric factors required to extract partial derivatives necessary for the ground-based navigation filtering process such as range to the spacecraft, observation time, spacecraft geocentric equatorial declination, and station location coordinates. As part of the modeling process, the appropriate measurement partial derivatives were derived. Since the Deep Space Network typically uses a measurement smoothing time of one minute, and since it is impractical to process the filter equations once a minute which would represent several million iterations for an outer planet mission, a model for compressing a number of Deep Space Network measurements into one was derived. This model approximates the state estimate for a set of measurements by a least squares estimate, and, assuming constant geometry over the sequence, allows a set of measurements and measurement partials to be replaced by a single value. The Deep Space Network modeling also includes station location biases and doppler noise as parameters.

The onboard navigation system is modeled, in Phase A, as a white noise system with a set of restrictions of measurement geometry including a selected real star field, restricted sun to optical line-of-sight angles, plane-spacecraft ranges, etc. For these studies, scanning photometer type instruments were the only types considered. For the Phase C studies the onboard navigation instrument model was extended to include an unestimated bias error. A TV type sensor was also considered.

The navigation simulation does not require that the guidance system be modeled in great detail, but only requires that the effects of guidance errors be accounted for. In the high thrust case, this amounts to modeling errors in the impulsive velocity increment magnitude. Direction information about the individual velocity corrections is lost because the noise sources are assumed to have equal probability of being in any direction. Consequently, the only guidance error sources of importance are those which affect the length of the velocity vector to first order.

STATISTICAL SIMULATION—LOW-THRUST CASE

Since for these missions a continuous thrust is used, the mid-course velocity correction is also a continuous process. Thus discrete velocity corrections were not incorporated as they were for the high thrust case. The extrapolation of statistics and the course corrections are combined in the solution of a set of differential equations describing the statistics. The input is the various correlation matrices for the initial injection or the terminal correlation matrices for a previous leg of the same mission. The number of "decision points" is preselected and at each decision point it is determined if a DSN or onboard measurement should be taken. If accelerometer measurements are taken during a leg, they are incorporated "continuously".

MODELING—LOW-THRUST MISSIONS

Three other programs were used besides the main program which performed the error analysis. The first was the program which created and stored the reference trajectory. Using this stored information the next program, the guidance program, created gain matrices which were stored. These matrices were used to get the perturbation on the control as a function of the state deviations. The third program generated cost matrices used for the measurement selection. Thus the nominal trajectory, gains matrices, and cost matrices were stored for each mission prior to the running of the main program. The navigational models used for the low-thrust mission simulations are essentially identical to those used in the high-thrust missions with the exception that accelerometer measurements are used here for navigation purposes. The incorporation of these measurements is done in the same general way as in the high-thrust case. There, a nine-dimensional state made up of spacecraft position and velocity and target planet position was used. In this case the state had the above nine components plus the spacecraft mass and two thrust vector misalignment angles. The onboard (except accelerometer) and DSN measurements do not directly measure these last three components of the state, so that the equations used for navigation were altered only so as to use the nine-dimensional state with zeroes added to the relevant matrices to make them dimensionally consistent.

The accelerometers which might be used on these low thrust missions provide either a continuous output or an output which is sampled with a period in the fraction of a second range. To process such data this frequently on these long missions would require a prohibitive amount of computer time. For this reason, a method of measurement compression was devised so that the effect of continuous accelerometer measurements over several days could be incorporated at one iteration as was done with the DSN measurements. The effect of a single accelerometer or three mutually perpendicular accelerometers were considered.

The low-thrust missions call for a more complex guidance model than the simple impulsive velocity input model used in the high-thrust case. For these missions a constant low-thrust engine which could be steered and turned on and off was used. The nominal trajectory which was a "minimum fuel" trajectory has associated with it a control history. If the real trajectory never deviated from this nominal, the thrust would have the direction and duration dictated by this control history. The object of the guidance scheme was to determine what the deviational control should be if the spacecraft deviates from the nominal path. The deviational or variational control could be either a change in the thrust direction or the switching of the thrust on or off at non-nominal times. Methods were considered for obtaining a change in the switch times as a function of extrapolated state vectors, but the guidance scheme finally utilized assumed that switches occurred at the nominal times and all guidance was by steering only.

The non-optimal scheme which was developed, has as its object the nulling of the components of the deviation in position which are perpendicular to the nominal velocity direction at the final time. The component of position in the direction of velocity cannot be controlled by steering alone. This component could, however, be diminished by altering the switch times if the last switch has not already occurred, or by changing the nominal arrival time. Other possible guidance systems are briefly discussed in Volume III.

HIGH THRUST GUIDANCE AND NAVIGATION RESULTS

GUIDANCE RESULTS

For these studies the guidance system is modeled as an impulsive thruster with an uncertainty occurring in the magnitude of the impulse due to errors in accelerometer integration and engine cutoff time. The guidance results, therefore, show the effects that the impulsive uncertainty has on position and velocity uncertainties at various points in the high thrust missions.

A sample of typical guidance results, drawn from the body of such data in Volume II, is the data from the Neptune passage of the Grand Tour mission presented in Table 1. The table shows the effects of accelerometer

Configuration	Periapsis Error Values	Terminal Error		
	RMS Position Estimate Error (km)	RMS Pos. Estimate Error (Km)	RMS Velocity Estimate Error (m/sec)	FTA Guidance Error (Km)
Nominal	12.65	102.03	1.38	113.9
Engine Cutoff Uncertainty X10	12.65	102.69	1.41	117.8
Engine Cutoff Uncertainty X100	12.01	143.30	2.66	323.6
Accelerometer Bias X10	12.65	154.11	2.92	254.1
Accelerometer Bias X100	12.75	216.84	4.92	5015.9
Accelerometer Scale Factor X10	12.65	102.24	1.39	114.4
Accelerometer Scale Factor X100	12.65	119.04	1.94	159.3

TABLE 1. Guidance Error Survey for Neptune on Grand Tour.

bias, scale factor error, and engine cutoff time uncertainty on the various position estimates at periplanet and at termination of the near planet leg of the mission when the probe reappears from behind Neptune. The significant guidance result from the table is that accelerometer bias uncertainty is the dominant error source. It appears that accelerometer bias as large as 0.1 cm/sec^2, along with scale factor errors up to 500 ppm are acceptable at the end of a nine-year mission. An unsolved problem in design for reliability exists in determining what instrument performance is required at earth departure such that the above values will exist after nine years.

Another guidance result comes from the guidance sensitivity matrix. Position and velocity error sensitivities are given as a function of time for the near planet passes. As an example, Fig. 1 shows the RSS velocity

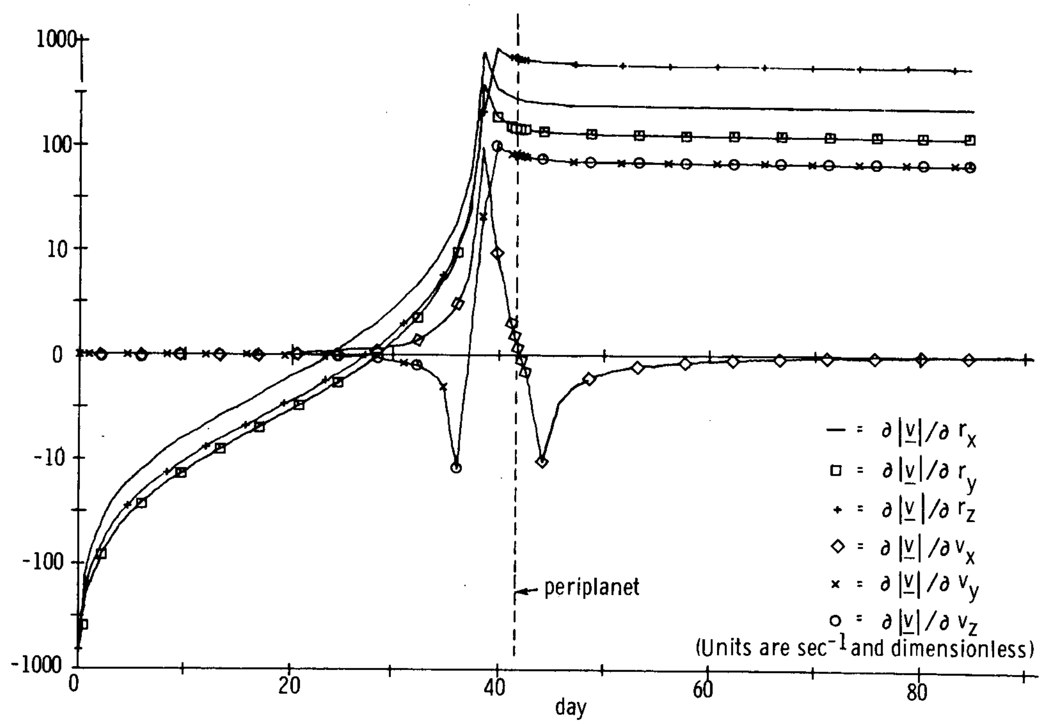


FIGURE 1. RSS Velocity Error Propagation Due to Initial Position and Velocity Errors for Neptune Flyby of 1977 Grand Tour.

error which would result at the time indicated by the abscissa, from a unit error in each component of position and velocity, as the probe makes its inbound intersection with the Neptune sphere of influence. The coordinates x,y,z are with respect to an ecliptic system with z perpendicular to the ecliptic and x along the ascending mode. Note the rapid increase of errors as periplanet is passed. The implication here is that it will be difficult to predict position and velocity through pericenter unless the input of new navigational information offsets this increase. Thus, without navigation measurements during this period, large guidance errors can ensue. The curves display only the effect of trajectory dynamics—no planet mass uncertainty which increases the errors was included.

NAVIGATION RESULTS

Jupiter Flyby

One of the prime objectives of these studies has been to examine the relative utility of onboard and ground-based navigation. By looking at the position and ephemeris errors at selected mission points it is possible to examine this question of relative utility. Tables 2 and 3 summarize the results for the interplanetary and near-planet legs of the 1973 Jupiter flyby mission. Under heading "configuration" at the top of the left-hand column are listed the various combinations of onboard and ground-based navigation considered. OB stands for onboard, DSN for Deep Space Network, and the pair of numbers indicate the assumed accuracy for visible and infrared onboard navigation instruments respectively.

The rms Position Estimate at sphere of influence arrival is given in the first column of Table 2. It can be seen that, with the nominal 10-arc-second visible light and 60-arc-second infrared light uncertainty chosen for the onboard instrument, the DSN navigation facility is vastly superior to onboard navigation. The combination of DSN and onboard result in a modest improvement of the overall position uncertainty. Reference to the second column of Table 2 indicates the refinement with which DSN can measure velocity as compared with the onboard system. Onboard-only errors are two-hundred times as great as DSN-only values. A combination of onboard with DSN does not enhance the velocity knowledge over that for DSN only. Turning to the column entitled rms Ephemeris Estimate, it is evident that on this interplanetary leg the onboard system is competitive in accuracy with DSN tracking, and that a combination of DSN with an onboard instrument yields the lowest values.

It is clear that the availability of the onboard system to reduce the ephemeris error improves as instrument quality gets better and that a reduction of errors over the DSN-alone capability is thus possible. By waiting until the probe is under the gravitational influence of Jupiter, however, we can reduce the ephemeris error even more by using ground tracking. This is evident from inspection of the ephemeris error column of Table 3. The column of Table 2, covering fixed-time-of-arrival guidance (FTA) error, refers to the actual position error of the spacecraft upon arrival at the Jovian sphere of influence. It is quite conclusive that reliance on an onboard system only results in a large terminal error and also in a substantial fuel penalty. The combination of onboard navigation with DSN produces an FTA error that is lower than DSN only, provided that a second velocity correction is made after the onboard navigation system has reduced the ephemeris error. The initial ΔV assumed for the launch vehicle in the fifth column of Table 2 is probably overly conservative in view of more recent launch vehicle analysis which shows these values to be about 18 mps.

Table 3 lists results similar to those of Table 2, but for the near planet portion of the mission, i.e., within the Jovian sphere of influence. Onboard-only navigation yields position uncertainties greater by a factor of ten than the DSN-only case. The use of the DSN and onboard in conjunction produces the minimum class of errors. In this situation the deletion of infrared sensing does not cause larger errors. The onboard system alone produces the largest velocity estimate errors, and the largest ephemeris errors. The velocity errors are larger simply because all the onboard measurement strategies considered here observe directly some component of position; none give directly any component of velocity. Ground tracking

Configuration	RMS Errors at Jupiter Sphere of Influence			Requirements		
	Position Estimate (Km)	Velocity Estimate (mps)	FTA (Km)	Ephemeris Estimate (Km)	Initial ΔV mps	Total ΔV mps
OB Only	9433.64	0.2101	10082.5	551.8	59.16	115.68
DSN Only	566.13	0.00085	559.7	555.1	52.29	55.40
DSN & OB 10''-60''	535.57	0.00082	930.7	534.5	52.29	55.07
DSN & OB 3''-60''	442.48	0.00082	928.4	441.1	52.29	55.07
DSN & OB 1''-60''	270.43	0.00085	471.5	268.1	52.29	55.20

TABLE 2. Results for Earth-Jupiter Interplanetary Leg of the 1973 Jupiter Flyby Mission.

Configuration	RMS Periapsis Estimation Errors		
	Position (Km)	Velocity (mps)	Ephemeris (Km)
OB Only	16.29	6.5051	551.5
DSN Only	2.75	2.6789	83.8
DSN & OB 10''-No IR	0.82	.7059	78.1
DSN & OB 3''-No IR	0.51	.3088	77.6
DSN & OB 1''-No IR	0.45	.2094	77.6
DSN & OB 10''-60''	0.80	.6844	76.4
DSN & OB 60''-60''	1.34	1.2578	79.0

TABLE 3. Near Planet Results for 1973 Jupiter Mission, Jupiter Passage.

on the other hand, provides an excellent observation of the component of velocity along the Earth-Spacecraft line. The ephemeris error is large in the onboard-only case because once the spacecraft is affected only by the gravity field of Jupiter there is no reference to any body but Jupiter; thus there is nothing with respect to which the location of Jupiter can be measured. Ground tracking, of course, always has Earth as a reference; hence once the probe is observed to be under the influence of Jupiter's gravity field, information on the location of Jupiter with respect to Earth can be determined. Comparison of onboard-only navigation with the DSN-only capability shows that in all respects ground tracking alone is superior to onboard navigation alone.

In general, with respect to navigation on the 1973 Jupiter mission, both onboard instruments and DSN tend to gain information only in the last ten days, when it is gained rapidly. During the last ten days, the onboard instrument tends to drop the uncertainties faster but not by much.

To summarize requirements for the 1973 Jupiter flyby, attention needs to be focused on the passage of the planet itself. DSN-only navigation as described in Table 3 yields uncertainties in position, velocity, and ephemeris values which are not excessive. The onboard-only case has large uncertainties and is unacceptable because of its poor performance on the interplanetary leg. The combination of onboard and DSN should only be considered if it is necessary to reduce the ephemeris error early in the encounter. This seems unlikely.

Jupiter Swingby to Saturn

Results for the interplanetary legs of this 1977 mission show that navigating on either interplanetary leg with an onboard navigation system alone, yields far poorer results than using Earth-based tracking alone. This is both in terms of navigational errors and fuel consumption. This poor performance of the onboard system is due to the extremely large distances to the nearest navigational targets encountered during the interplanetary phases of this mission. The Earth-Jupiter leg of this mission produces much the same results with regard to ephemeris error reduction as were observed on the interplanetary leg of the Earth-Jupiter mission. The onboard instrument, if it is very accurate, can reduce ephemeris errors considerably although it is limited by the value of position error which the DSN provides. This pattern repeats again on the approach to Saturn on the Jupiter-Saturn leg of this mission. The percentage reduction in error is, however, greater on the Saturn approach than on the Jupiter approach. The reason for this is that the ephemeris error for Saturn is larger than that of Jupiter, hence the onboard system can be effective sooner. The results show, however, that by waiting until the spacecraft is within the sphere of influence of either of these planets the ephemeris error can be reduced still further even with only DSN tracking. Thus it may be concluded, that unless there is some need to reduce the ephemeris error early in the encounter, there will be no requirement to use an onboard system on either interplanetary leg of this mission.

The major difference between the Jupiter passage results on this mission and on the similar passage of the Jupiter Flyby mission is that onboard-only navigation is competitive with DSN-only navigation when comparing errors at periplanet. The blending of information from ground tracking and onboard systems gives a noticeable improvement over each individually if periplanet errors are compared. This same improvement, when using both, is apparent in the terminal errors also but a very precise visible spectrum instrument is required to obtain it. The reason for this favorable comparison with the Jupiter Flyby is not that the onboard system works better in this case, but that ground tracking is less effective because of the shorter time spent inside the sphere of influence on this higher energy mission.

This small reduction of errors obtainable by adding the onboard capability to the ground tracking capability seems to be the main benefit from using an onboard system on the first two legs of this mission. There is no noticeable fuel saving gained and the early reduction of the ephemeris error probably does not justify the addition of the extra navigation capability.

Turning now to the Saturn Flyby summarized in Table 4, we note that the very small navigational errors characteristic of every periplanet

Configuration	Periapsis Position Estimate (km)	Position (km)	Velocity (mps)	Ephemeris (km)
OB only	86.34	20.31	3.19	627.2
DSN only	48.20	16.38	5.78	257.3
DSN & OB 10" - No. IR	13.81	7.56	0.64	255.7
DSN & OB 3" - No. IR	10.03	6.18	0.39	255.6
DSN & OB 1" - No. IR	8.03	5.78	0.31	254.9
DSN & OB 10" - 60"	14.76	7.89	1.50	253.0
DSN & OB 60" - 60"	28.97	9.19	0.56	256.2

TABLE 4. Near Planet Results for 1977 Saturn Mission, Saturn Passage.

point examined thus far are prevelant. The use of ground tracking during the close passage provides a means of substantially reducing the ephemeris error for Saturn. The use of an onboard system adds nothing to the ability to learn about the planetary ephemeris. This is because the location of Saturn (or any other planet) with respect to the Earth or Sun is not observable with an onboard system unless it sights on one of these two bodies. Unfortunately, they are so distant that the measurement is too noisy to be useful for navigation measurements.

The possibility of producing a net system-weight saving by using onboard navigation to reduce ΔV requirements was investigated, and the results show that only if the spacecraft initial mass is greater than about 1500 kg can a saving be effected.

Grand Tour

As with the previous missions, the results for the 1977 Grand Tour mission show that the performance on interplanetary legs with only an onboard navigation system is substantially inferior in determining vehicle location than with the Deep Space Network alone. However, at the Saturn passage on this Grand Tour mission, the balance between the onboard system and the DSN changes from that observed at Jupiter. Using onboard navigation only now results in both smaller position errors and smaller fuel requirements than tracking from Earth without onboard augmentation. Combining the two systems results in still smaller errors and fuel requirements. The same pattern of results exists with even larger fuel savings at Uranus and stronger reduction of errors at Neptune by adding the onboard system. Thus it may be concluded that there is a substantial reduction in midcourse fuel requirements and navigtional errors if an onboard navigation system is employed on the Grand Tour for use in the approaches and encounters at Saturn, Uranus, and Neptune.

The Saturn passage is the first instance of a case where onboard-only capability proves to be better than DSN-only. The response to the combination of both navigation methods is, however, the strongest of all, both in terms of minimizing errors at passage and in terms of fuel consumption. Also, the errors do not respond strongly either to infrared capability or to enhanced accuracy. Remembering that this trajectory goes between Saturn and its rings, guidance errors must be minimized. Even the 60" - 60" system will reduce the out of path position error component from 326 km to 31 km, a factor of ten. With the addition of a final velocity correction, the onboard 60" - 60" capability reduces the out-of-path value from the DSN-only value of 106 km to 5 km. The onboard system is not justifiable in terms of significantly reduced uncertainties at exit from Saturn's sphere of influence enroute to Uranus. However, the same size state vector uncertainties and guidance errors are achieved with significantly less fuel. The use of the 60" - 60" configuration reduces the total velocity change requirement from 123 meters per second to 42 meters per second.

In order to pursue the subject of tradeoff between DSN-only and DSN-and-onboard capability further, a comparison was made using spacecraft with assumed initial weights. The results of this study are shown in Table 5. Velocity totals for each of the eight mission legs of the Grand Tour were chosen, for the DSN-only and the DSN with OB (1" - 60") cases. The three assumed spacecraft weights at earth departure were

MISSION PHASE	Velocity Budget (mps)		Final Mass (kg) at End of Mission Phases					
			W_o 226.8 (kg)		W_o = 453.6 (kg)		W_o = 2268.0 (kg)	
	DSN Only	DSN&OB 1" - 60"	DSN Only	DSN&OB 1" - 60"	DSN Only	DSN&OB 1" - 60"	DSN Only	DSN+OB 1" - 60"
Earth-Jupiter	56.0	56.1	220.2	220.2	440.5	440.4	2202.4	2202.1
Jupiter Passage	9.47	2.60	219.2	219.9	438.4	439.8	2218.9	2199.2
Jupiter-Saturn	1.1	2.4	219.0	219.7	438.1	439.3	2190.5	2196.6
Saturn-Passage	134.33	11.28	203.5	218.4	407.1	436.8	2035.4	2183.9
Saturn-Uranus	4.6	6.13	203.1	217.7	406.1	435.4	2030.5	2177.1
Uranus Passage	252.80	9.29	175.1	216.7	350.3	433.4	1751.5	2166.8
Uranus-Neptune	9.0	9.05	174.4	215.7	348.7	431.4	1743.5	2156.7
Neptune Passage	9.1	15.85	173.5	213.9	347.0	427.9	1735.3	2139.3
Mass Savings-kg (Gross)				40.4		80.8		404.0

TABLE 5. Grand Tour Spacecraft Weight Estimates.

chosen 227 kg, 454 kg, and 2268 kg (500, 1000, 5000 lb). The resulting gross weight savings using DSN with OB are given at the bottom of the table and indicate a compelling reason to use onboard navigation on the Grand Tour.

The gross weight savings in these cases are roughly proportional to the spacecraft gross weight (18%). Of course the net weight savings involves the direct and indirect weight and reliability penalties incurred by actually adding the equipment. The onboard navigation sensor mass has been estimated at from 4 to 20 kg with the latter number including 8 kg of onboard computer and assuming a four-degree-of-freedom sextant instrument.

Comet Rendezvous Missions

Two missions to short period comets via a Jupiter swingby were analyzed. For both missions it was found that the DSN Earth tracking provided adequate navigation accuracy during the Jupiter passage so that navigation near rendezvous was unaffected by the Jupiter navigation uncertainties. A TV sensor was chosen as the nominal onboard navigation sensor for the comet missions because the weak comet radiation level appears to require a device with signal integration capability.

Comparisons between the two missions are listed in Table 6. The more important differences include the terminal closing speed which affects the time available for navigation measurements, the ephemeris uncertainty which affects the initial ΔV and the navigation sensor search problem, and the Earth-Comet distance at rendezvous which is inversely proportional to the Earth sighting uncertainty arc length. Comparing numbers in Table 6 shows that the P/Tuttle-Giacobini-Kresak mission is generally more difficult with the exception of a smaller Earth-comet distance at rendezvous.

Nominal assumed simulation parameters are listed in Table 7. Earth telescope sighting accuracy is set at $0.3''$ which is a median value between the best star accuracy of $0.03''$ and an average planetary sighting of $3''$. The radial ephemeris error component is along the Earth-comet line of sight. The onboard navigation sensor is assumed to be an image tube with $2''$ white noise and a $5''$ bias error. Onboard navigation sensor turn-on time is much earlier for the P/Tempel 2 mission because of the lower closing speed and the fact that P/Tempel 2 is a considerably brighter object. The second ΔV is assumed to occur after some onboard system sightings have been made.

Nominal case results for the two missions are listed in Table 8 where terminal errors are specific to conditions one hour before the final rendezvous burn; thus, these results are also applicable to slow flyby missions. Position uncertainty with respect to the comet is seen to be smaller for the P/Tuttle-Giacobini-Kresak mission, and this reflects one of the key characteristics of these missions, namely that the Earth-based sighting accuracy ultimately limits the navigation accuracy. This is because the onboard system gathers no effective range information, so that the spacecraft-comet line of sight uncertainty is bounded by the Earth sighting input. The components of position uncertainty normal to the line of sight are by contrast very small. Guidance position error is roughly the same as the navigation uncertainty since the last course correction is limited

Table 6 Comet Mission Comparisons		
Quantity	Tempel 2	P/Tuttle-Giacobini-Kresak
Total flight time	1591 days	1690 days
Jupiter passage distance	24.9 Jup. radii	204.8 Jup. radii
Arrival time (days before perhelion)	58	50
Terminal closing speed	3.6 km/sec	5.7 km/sec
Ephemeris uncert. (largest component)	4000 km	500,000 km
Earth-comet dist. (at rendezvous)	2.7 a.u.	1.2 a.u.
Comet nuclear radius[3]	0.6 km	0.1 km

Table 7 Nominal Comet Mission Parameters		
Parameter	Tempel 2	P/Tuttle-Giacobini-Kresak
Earth telescope accuracy	$0.3''$ (600 km)	$0.3''$ (260 km)
Ephemeris uncert. (radial value)	400 km	10,000 km
Image tube noise (white noise, bias)	$2''$, $5''$	$2''$, $5''$
Onboard nav. turn-on time	E-10 days	E-2 days
Earth sighting input time	E-20 days	E-20 days
Time of velocity corrections	E-19, 8 days	E-19, 1.5 days

Table 8 Nominal Comet Mission Results		
Quantity	Tempel 2	P/Tuttle-Giacobini-Kresak
Position uncert. mag. (respect to comet)	608 km	285 km
Line of sight comp.	608 km	285 km
Normal components	3.5 km	2.6 km
Guidance position error	616 km	286 km
1st velocity correction	3.88 m/sec	298 m/sec
2nd velocity correction	1.26 m/sec	79.5 m/sec

by navigation precision. The initial velocity correction for the P/Tempel 2 mission is smaller by nearly two orders of magnitude, and is proportional to the initial ephemeris uncertainty, while the second correction is smaller primarily because it is made earlier on this mission (8 days) than on the P/Tuttle-Giacobini-Kresak mission (1.5 days). The P/Tuttle-Giacobini-Kresak ΔV's are of such magnitude that they would most likely be operationally incorporated with multiple rendezvous ΔV applications, to reduce the effect on the total ΔV carried by the S/C.

Jupiter Entry Missions

Two 1978 Jupiter entry missions of 800 and 1200 day duration were simulated. For these missions the quantities of primary interest were the entry conditions including the radial distance from the mass centroid, the entry angle, the entry velocity, and the geographical location of the entry point.

The two missions differ mainly in the location of the entry point and the associated time of flight. The 800 day mission approaches from the sunlit side and enters near the terminator. The 1200 day mission approaches from the dark side and enters near the sub-sun location.

Comparisons of nominal case results for the two missions are given in Table 9. Using an onboard instrument of 5 arcsecond accuracy reduces

Table 9 Comparison of 800 Jupiter and 1200 Day Missions

	800Day	1200 Day
Term pos est error - DSN	3	14
-DSN + OB	1	1.2
Term guid pos error - DSN	680	810
-DSN + OB	357	306
Entry angle est error - DSN	$.00037^{o}$	$.00091^{o}$
-DSN + OB	$.00022^{o}$	$.00026^{o}$
Entry angle guid error - DSN	$.26^{o}$	$.32^{o}$
-DSN + OB	$.14^{o}$	$.12^{o}$

the position estimation uncertainty from that achieved by the DSN, but the improvement is in a value that is already small. For example, there is a reduction from 3 to 1 km in the 800 day case. Terminal guidance position error is seen from the table to be more strongly affected by the addition of onboard navigation with a reduction by a factor of two when the final ΔV is at E-3 days. The entry angle guidance and estimation errors are seen to be only moderately affected by the addition of an onboard sensor. The tabulated values for the nominal cases are expected to be acceptable errors and uncertainties for the 15^{o} entry angle considered, however these values can be seriously degraded if the DSN measurements are restricted within a certain terminal period. Such a restriction might arrise when communications requirements result in a flyby bus being separated from the entry probe which is on a direct entry trajectory. Under these conditions the entry angle guidance error would degrade as in Fig. 2. where the last

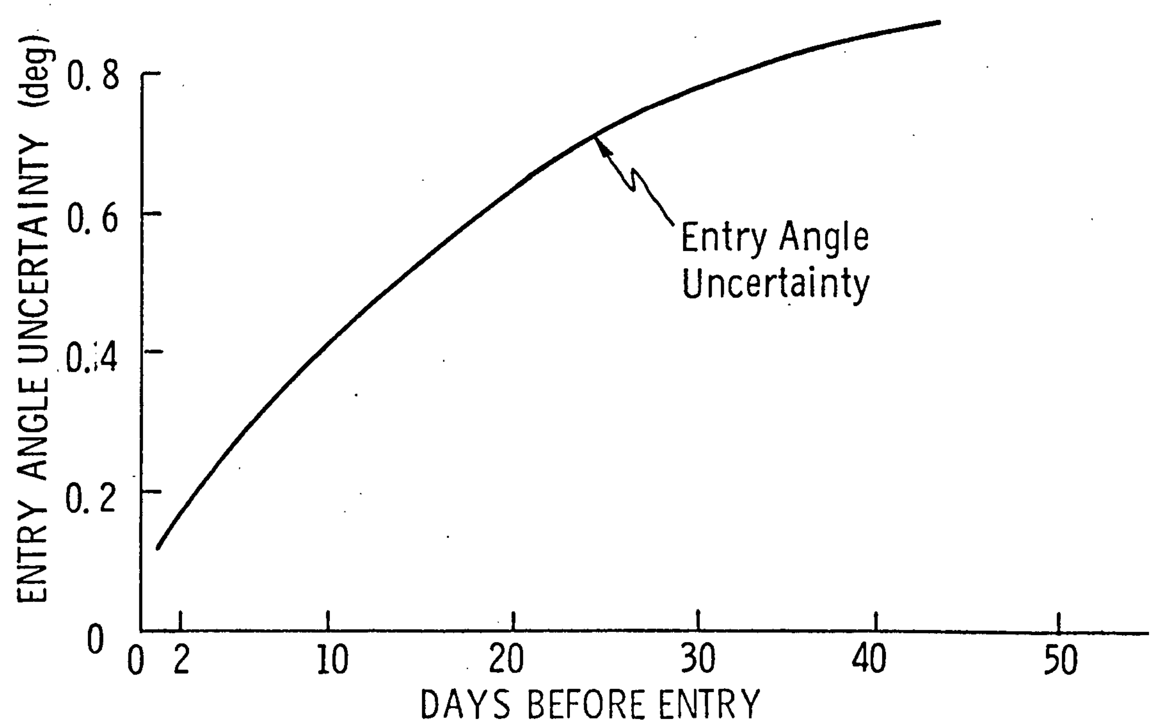


Figure 2 . Effects of Early DSN Shutdown on Terminal Entry Angle Error

midcourse correction is made one day after DSN shutdown. Nominal ΔV requirements for the entry missions are 4.4 m/s and 18.3 m/s for the 800 and 1200 day missions respectively (excluding the post-Earth injection error correction of 25.2 m/s and 47.6 m/s respectively).

Multiple Planet Swingby Missions

The positions of the planets Uranus, Neptune and Pluto are such that for the next several decades an assortment of three-body Grand Tours are possible using Jupiter as the first of the sequence. The two missions considered here are the sequence Jupiter-Uranus-Neptune, to be launched in 1979, and Jupiter-Saturn-Pluto to be launched in 1977.

The three planet swingby studies are an extension of Phase A work which examined a two planet swingby and the four planet Grand Tour. The present studies sought to determine whether or not the relative utility of onboard navigation is altered when onboard sensor process noise is added to the model, and whether a TV type sensor making sightings on the planetary satellites is an effective navigation device. TV is considered in addition to and not in place of the nominal scanning photometer.

The results show that onboard navigation remains extremely useful for navigation error reduction on these missions. For example Fig. 3 shows the position uncertainties at the inbound and outbound sphere of influence and at pericenter for the Jupiter-Uranus-Neptune mission. Values are shown for both DSN and DSN with onboard cases, and it is clear that onboard navigation reduces position uncertainties considerably even though the onboard sensor is assumed to have a 5 arcsecond unestimated bias error. The effect is particularly pronounced at pericenter.

Onboard navigation cases using a scanning sensor and an image tube type sensor are compared in Table 10 for the Saturn passage on the Jupiter-Saturn-Pluto three planet swingby.

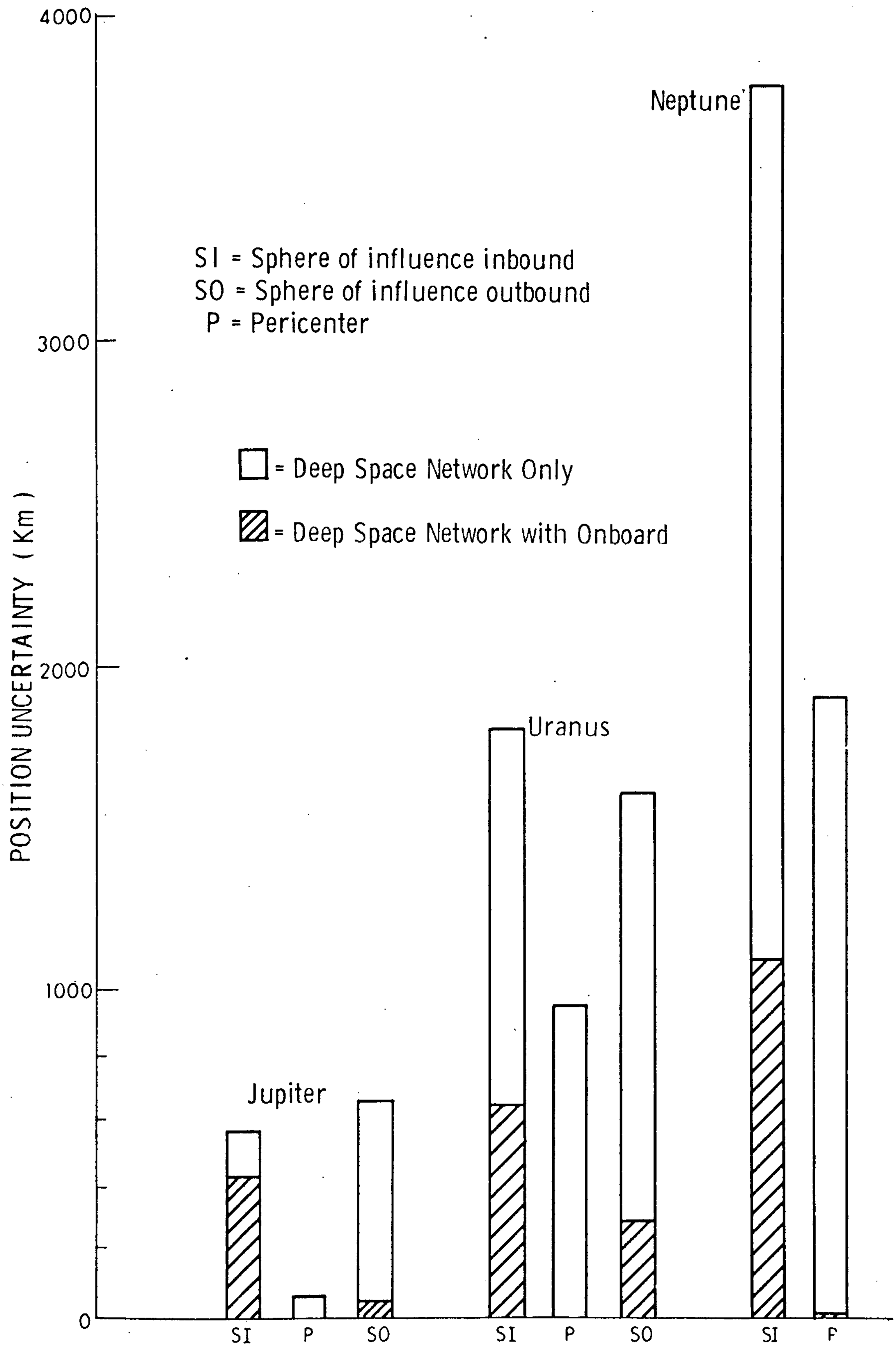


Fig. 3 Neptune 3 Planet Swingby Position Uncertainty

Table 10. TV Navigation Saturn Passage

Quantity	DSN only	1σ Satellite Ephemeris Uncert. (km) 9000	2700	900	Scanning Photometer
1σ Pericenter Position Uncert. (km)	475	126	126	93.5	30.3
1σ Position Uncert. (km) Sphere of Influence Outbound	1686	1651	1650	1412	239

Position uncertainties are compared at pericenter and at the outbound sphere of influence. The uncertainty in image tube sightings on the outer planet satellites depends upon the initial satellite ephemeris uncertainty, and this quantity was parameterized by factors of three and ten from the assumed nominal value of 900 km. In both cases the TV sensor makes an improvement on DSN only values, and variations in the initial ephemeris are relatively unimportant. At pericenter the scanning sensor yields lower uncertainties because it is allowed to continue to gather information right up to pericenter whereas the TV sensor is turned off at 50 planetary radii. On the outbound leg TV is relatively ineffective since the satellite, Titan, upon which it would be sighting, is too near the sun to make sightings over much of the outbound, near-planet leg. The scanning photometer, sighting on Saturn, does not have this problem and consequently produces a smaller uncertainty. Velocity corrections for the multiple planet swingby missions are listed in Table 11.

TABLE 11

Nominal ΔV Summary for Multiple Planet Swingbys

Neptune Mission[1]

	Time of ΔV_1	ΔV_1 Mag. (M/sec)	Time of ΔV_2	ΔV_2 Mag. (M/sec)	Time of ΔV_3	ΔV_3 Mag. (M/sec)	Time of ΔV_4	ΔV_4 Mag. (M/sec)
Jupiter Passage	E-44.82	1.24	E-2	1.74	E+56.82	0.67		
Uranus Passage	E-38.64	1.96	E-2	2.75	E+2	15.1	E 50.64	10.68
Neptune Passage	E-50.11	2.52	E-2	4.45				

Pluto Mission[2]

	Time of ΔV_1	ΔV_1 Mag. (M/sec)	Time of ΔV_2	ΔV_2 Mag. (M/sec)	Time of ΔV_3	ΔV_3 Mag. (M/sec)
Jupiter Passage	E-44.98	1.20	E-6.98	0.43	E+47.98	1.05
Saturn Passage	E-38.54	0.92	E-2	5.04	E+50.54	1.15
Pluto Passage	E-7.50	30.6	E-2	22.5		

[1]does not include injection error correction of 25.6 M/sec.

[2]does not include injection error correction of 25.9 M/sec.

LOW THRUST GUIDANCE AND NAVIGATION RESULTS

GUIDANCE RESULTS

Because of numerical problems, the suboptimal, constant control guidance scheme described above under the heading of low thrust guidance modeling, was not coupled to the statistical simulation. The net result of this is that the guidance and navigation results were uncoupled in the statistical simulation. However, a number of results can be obtained by examining the characteristics of the guidance equations. Using the gain matrix, and given a maximum allowable deviation of thrust direction from nominal, one can determine the maximum nullable deviation of the state vector from the nominal at any given time. Similarly, given a maximum allowable state deviation one can work back and find the required off-nominal thrust control angle. These calculations have been performed for the low thrust missions and are displayed in a set of curves in Volume III of this report.

As a partial example of the use of these data, Fig. 4 and 5 will be used to compute the control perturbation and maximum permissible state deviation for a typical Saturn orbiter.

Considering this Saturn near planet case and a time to go of 40 days, assume that a deviation exists in the y component of velocity which is equal to 0.02 km/sec. From Fig. 4 the in-plane control deviation, $\Delta\theta$, is given by the product of the inverse value of the curve at 40 days and $\Delta V_y = 0.02$,

$$\Delta\theta = \frac{1}{.01} (.02) = 2^{\circ}$$

From Fig. 5 the out of plane control deviation, $\Delta\psi$, is given by

$$\Delta\psi \frac{1}{.3} (.02) = .067^{\circ}$$

Thus if

$$\Delta V_y = 0.02 \text{ km/sec}$$

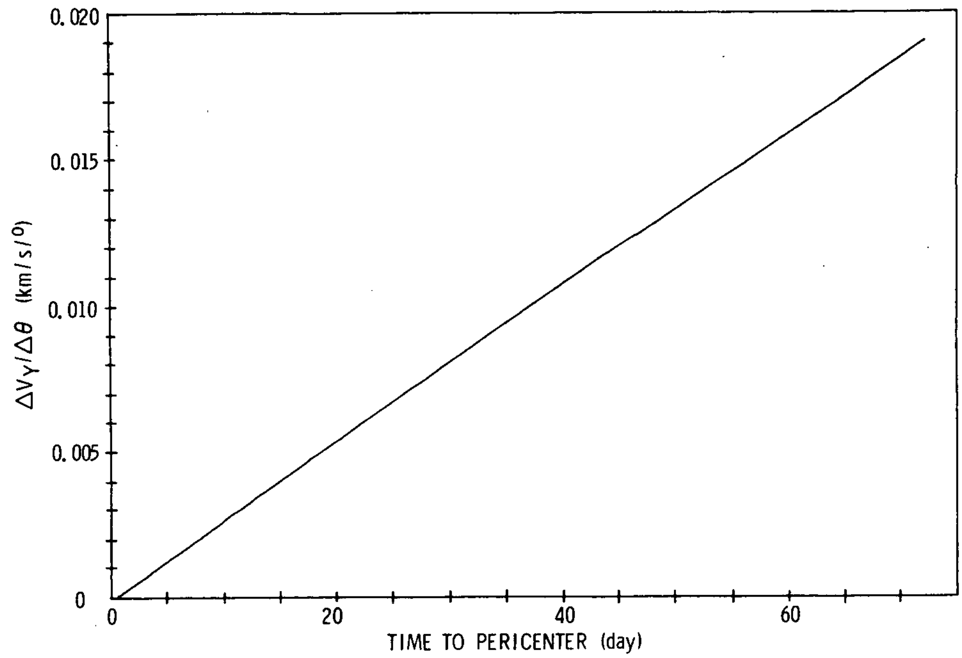


FIG. 4 Y-Component of Velocity Deviation Per In Plane Control Angle Deviation vs Time to Go at Saturn

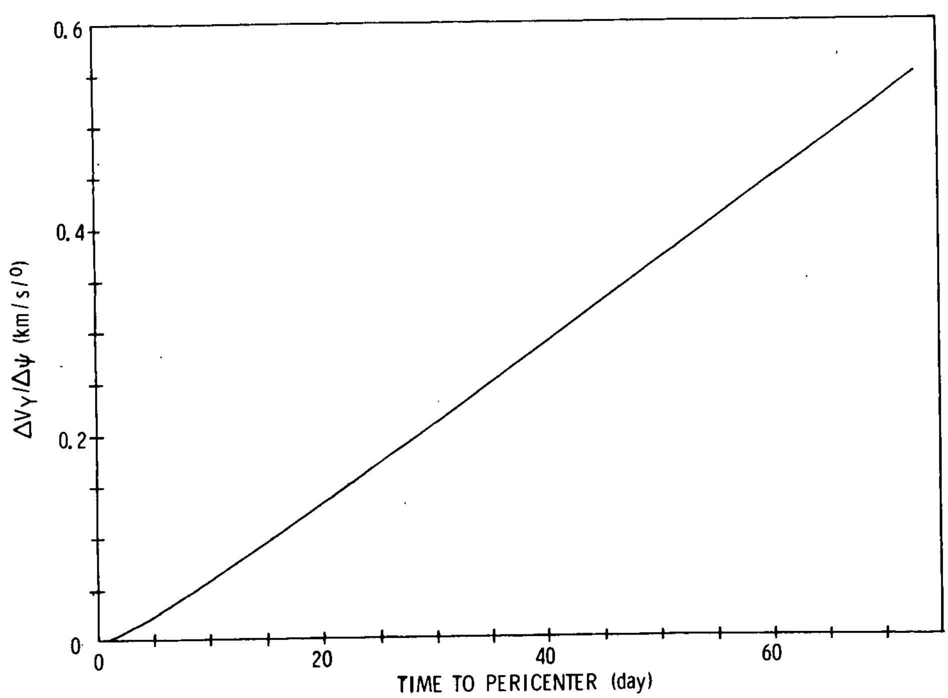


FIG. 5 Y-Component of Velocity Deviation per Out of Plane Control Angle vs Time to Go at Saturn

then

$$\Delta\theta = 2^\circ$$

$$\Delta\psi = 0.067^\circ$$

Now say that there is a limit imposed on the magnitude of $\Delta\theta$ and $\Delta\psi$ of 1°. Since 2° exceeds the 1° limit the $\Delta V_y = 0.02$ km/sec cannot be nulled. Using Fig. 4 and 5 the greatest value of ΔV_y that can be nulled can be found. The product of the value of the curve at 40 days in Fig. 4 and $\Delta\theta = 1^\circ$ is

$$(0.0105)(1^\circ) = 0.01 \text{ km/sec}$$

The product of the value of the curve at 40 days in Fig. 3 and $\Delta\psi = 1^\circ$ is

$$(0.3)\ (1^\circ) = 0.3 \text{ km/sec}$$

Thus the limit imposed on $\Delta\psi$ would allow a deviation $\Delta V_y = 0.3$ but the limit on $\Delta\theta$ allows only $\Delta V_y = 0.01$ km/sec. Thus in order for the deviation in V_y to be nulled, $|\Delta V_y| < 0.01$ km/sec. That is $|\Delta\theta| < 1^\circ$, $|\Delta\psi| < 1^\circ$ implies maximum $|\Delta V_y| < 0.01$ km/sec.

At the beginning of this example we hypothesized a $\Delta V_y = 0.02$. This is greater than the maximum nullable value if $|\Delta\theta| < 1^\circ$, $|\Delta\psi| < 1^\circ$. The maximum nullable $\Delta V_y = 0.01$. The excess is then

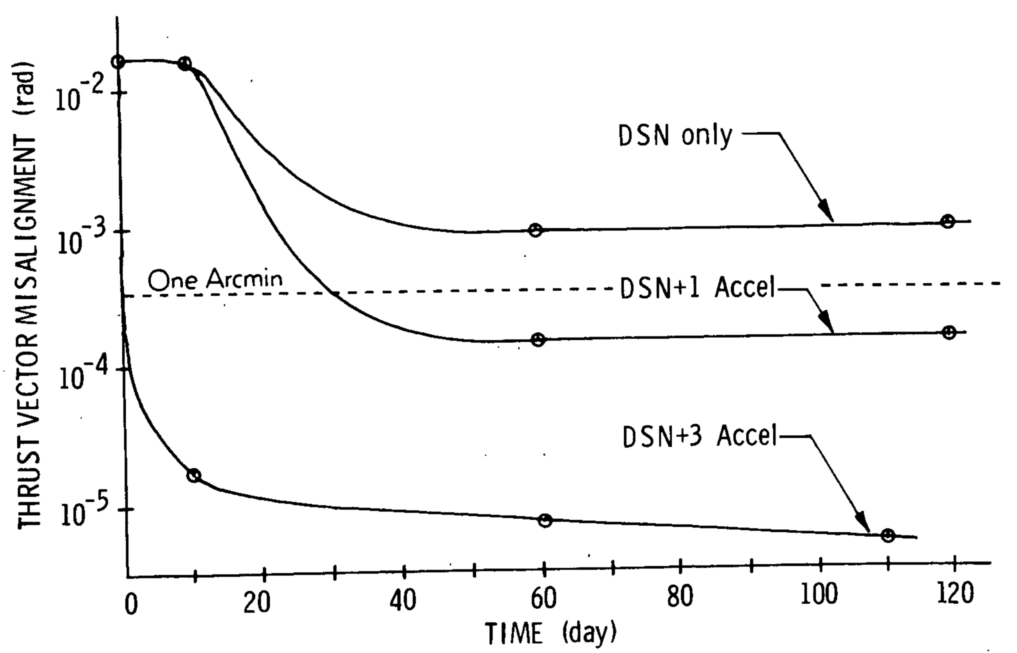
$$0.02 - 0.01 = 0.01 \text{ km/sec}$$

Using an additional set of curves from Volume III and this excess value of velocity deviation, one could then find the final position deviations which would result.

NAVIGATION RESULTS

The low-thrust mission results bear a great deal of similarity to the high-thrust Jupiter flyby results particularly with regard to the utility of the onboard navigation sensor. However, because of the possibility of accurately controlling thrust vectoring early in the mission, when the DSN is highly accurate, the addition of highly sensitive accelerometers is studied as a navigation parameter. The results for low-thrust missions will therefore show the effects of onboard sensor systems with various combinations of onboard sensor accuracies. Systems were studied that combined DSN with one accelerometer only, three accelerometers (orthogonally mounted) only, onboard navigation only, and onboard navigation combined with accelerometers. Figure 6 shows the speed with which accelerometers can drive down thrust vectoring errors on a Jupiter mission compared to the DSN-only system. DSN settles down to about 3 arc minutes, DSN with two accelerometers down at about 1/2 arc minute. DSN with

FIG. 6 Accelerometer Effect on Thrust Vector Misalignment



three accelerometers almost immediately drives the errors to a low value around a few arc seconds. Because of numerical limitations, DSN measurements were not allowed to decrease thrust vector misalignment beyond the approximate 3 arc-minute level although this limitation did not affect accelerometer measurements. This does correspond to expected limitations on the ability of the DSN to measure thrust vector misalignment.

Table 12 summarizes the effects of adding one accelerometer, nominally aligned with the thrust vector, to the system. The position errors at SOI are relative to the Earth not to Jupiter as in the high thrust cases. On the near planet leg of the Jupiter mission one sees a reduction of an already small position error, and a 20% reduction in ephemeris. On the Jupiter interplanetary part of the mission there are only slight reductions. From these numbers it can be concluded that one accelerometer has little utility on a Jupiter low-thrust mission (for Saturn the result is similar). Table 13 shows the same type of information for a three accelerometer system. On the near planet leg there is a large, but probably not significant reduction of position error from 12.9 km to 2 km, and a marked decrease in ephemeris error. This is the most significant error reduction caused by an onboard sensor in these studies. On the Jupiter interplanetary leg, the three accelerometers cut position error in half but have an insignificant effect on ephemeris error. Again the Saturn mission results are similar.

TABLE 12. 1 Accelerometer Utility for Error Reduction

	1 σ Position Error (Km)		1 σ Ephemeris Error (Km)	
	With no Accel	With 1 Accel	With no Accel	With 1 Accel
Jupiter Near Planet With OBN	3.0	0.68	282	236
Jupiter Interplanetary	149	145	490	490

	1 σ Position Error (Km)		1 σ Ephemeris Error (Km)	
	With no Accel	With 3 Accel	With no Accel	With 3 Accel
Jupiter Near Planet Nominal DSN	12.9	2.0	305	79.5
Jupiter Interplanetary Nominal DSN	149	64	504	482

TABLE 13 3 Accelerometer Utility for Error Reduction

Table 14 summarizes the low-thrust results related to the onboard navigation sensor. On the Jupiter mission, near planet leg, the navigation sensor makes further reductions in errors that have already been markedly improved by the addition of three accelerometers. If the DSN doppler noise level is assumed to be 100 times nominal, the onboard instrument is able to keep position errors small. The only other significant onboard navigation effect is the reduction in ephemeris error from 504 km to 220 km seen on the Jupiter interplanetary leg. This result depends on a one-arc second sensor capability.

	1 σ Position Error (Km)		1 σ Ephemeris Error (Km)	
	Without Onboard	With Onboard	Without Onboard	With Onboard
Jupiter, Near Planet 3 Accelerometers	2.0	0.3	79.5	50.0
Jupiter, Near Planet Doppler Noise x 100	32.6	2.3	388	368
Saturn, Near Planet 1 Accelerometer	6.8	6.3	490	490
Jupiter, Interplanetary	149	136	504	222

TABLE 14 Onboard Navigation Utility for Error Reduction

CONCLUSIONS AND RECOMMENDATIONS

HIGH-THRUST MISSIONS

Conclusions

For the 1973 Jupiter Flyby, the study concludes that onboard navigation can make no contribution to reducing the navigation error. The fixed-time-of-arrival (FTA) error was found to be very sensitive to accelerometer bias and propulsion cutoff uncertainty. A bias level below 0.10 cm/sec^2 and a cutoff uncertainty of 0.05 milliseconds should be maintained.

For the 1977 Jupiter Swingby to Saturn, onboard navigation did not make a significant impact upon the navigation accuracy until the actual Saturn passage at three radii was being approached. At this point, the fixed-time-of-arrival guidance error of some 480 km was reduced an order of magnitude by onboard navigation system measurements, combined with DSN navigation. There is some question whether or not this is a needed benefit, since at three Saturn radii the position vector at passage may not be a critical mission parameter. In any event, it was shown that, on a total system basis, the fuel savings produced by the potentially more accurate onboard navigation did not pay in weight for the additional weight attributable to the onboard instrumentation.

In the analysis of the 1977 Grand Tour the pattern for the Saturn mission was found to repeat itself in that the onboard navigation system begins to pay for itself at the Saturn passage. The combination of onboard navigation capability with DSN was found to result in both lower errors and fuel savings. For the complete mission, the savings in required onboard fuel for a given initial spacecraft weight amounted to about 18% of initial weight, enough to justify the penalty of the navigation sensor.

The guidance requirements were found to emphasize the same factors as in the 1973 Jupiter mission, namely accelerometer bias and cutoff time uncertainty. At the outer planets, the FTA guidance sensitivity to accelerometer bias and thrust cutoff errors did not appear until the errors had been increased generally by more than a factor of ten from the selected nominal values of .3 cm/sec and .05 sec.

The attitude control requirements were shown to be involved in a tradeoff with the navigation instrument. The simplest body-fixed sensor required complete maneuver and precision hold from the attitude control system. Navigation sensors of increasing weight, power, and complexity were shown to require less and less performance from the attitude control system.

In the Phase C extended mission studies the addition of an unestimated bias error to the onboard navigation sensor was found not to significantly change directly comparable results that were produced in Volume II of this series. Another additional noise source, namely the solar mass uncertainty, was examined, and it was found that increasing this uncertainty by a factor of 10 over the nominal value did not substantially alter the results.

Onboard navigation was found to be useful, under certain conditions, for each of the three classes of extended missions. It was found to be essential for the comet missions, very valuable for the multiple planet swingbys, and useful under very shallow entry angles for the Jupiter entry missions. Results for the comet missions show that Earth-based telescope sightings ultimately limit the navigational accuracy and scale the ΔV magnitudes. Relative position uncertainty of the comet with respect to the spacecraft is essentially given by the residual Earth-based uncertainty which is unsighted by the onboard sensor. Because of this limiting accuracy, the onboard sensor does not have to meet stringent accuracy requirements, and the nominal 0.3 arc second Earth-based sighting requires only arc minute-like onboard accuracy for compatibility. The two comets considered for these missions are extremely dim objects. Thus an onboard sensor that can integrate signal (like an image tube) appears to be a prime candidate for the task of comet detection.

Results for the Jupiter entry missions are focused upon the errors and uncertainties in the terminal conditions at the nominal entry point. It was found that the timing of the last navigation measurement is of primary importance, and that at about E-2 days the entry condition errors begin to increase rapidly with earlier navigation shutdown. In general, no onboard sensor was required for this mission however. If a very shallow entry angle was of interest, a limb sensor would be important for establishing the direction to the limb and thus for controlling the entry angle.

The results for the 3 planet swingby missions show that onboard navigation remains extremely useful in spite of the added onboard sensor bias error. Periplanet position uncertainties are reduced drastically by the onboard system, and ΔV's at the outermost planets are reduced considerably.

Navigation measurements made on the angle between a star and the near planet using a scanning device appear to be closely comparable to those made with a image tube measuring planetary satellite-star angles. The scanning device is, however, dependent upon DSN to some degree in the modeled mode of operation and the image tube finds an absence of satellites at Pluto. Both devices also can serve as valuable science instruments.

Recommendations

On the planetary passages position uncertainties are ultimately limited by the planetary navigation phenomenon uncertainty. Minimization of phenomena uncertainties will require a considerable amount of work beginning now and continuing through the outer planet flights. Data on all the pertinent experiments performed to date needs to be gathered and correlated. Further experiments that will complement the data on hand need to be designed and performed. Models of the radiative transfer processes need to be developed so that mathematical simulations can be constructed. It is anticipated that simulations will help separate useful from spurious experimental data and will provide the best possible horizon profiles prior to flight.

Tangential ephemeris errors might possibly be reduced by star occultation experiments, or error analyses applied to planetary satellite positions. The effects of latitude and cloud thickness and composition on radius uncertainties should be determined. Models of phenomena in the meteorological sense should be formulated so that the variations in horizon profiles can be estimated. Finally, the error models associated with the combining of instrument and phenomena errors should be expanded to include the results of multiple measurement filtering.

In the area of navigation instrument design studies there is a particular challenge. Phenomena physics and instrument design must be matched to each other. Yet, the phenomena physics will, in this case, be derived from the use of the instrument. Secondly, whereas inertial sensor and computer subsystems are under development which have survivability or graceful degradation, the navigation sensor subsystem stands out as lacking a conceptual approach to mission reliability. Intensive design work is needed now to feed into an instrument developmental program. This instrument should be flown and operated on a planned inner solar system planetary mission, in the early 1970's, so that operational experience can be gained.

It is felt that the guidance problem reduces to one of velocity correction vector control, and reliability. In this area, development programs should be aimed at single specific-force sensors for the outer planet mission environments. Better control and prediction of variation in the propulsion system "tail-off" impulse is a requirement.

This study was made without a series of candidate or representative spacecraft designs. However, the need was seen for a central digital computer to give the spacecraft mission operational autonomy and flexibility. The onboard measurements for the Grand Tour will require this support just for the operation of the instrument in the measuring cycle. With some definition of candidate spacecraft, it would be possible to make more satisfactory tradeoffs between instrument orientation flexibility and precision and spacecraft orientation flexibility and precision.

LOW-THRUST MISSIONS

Conclusions

From the standpoint of the navigation results, the onboard navigation system proves to be of little value except in two particular situations. First, if the Deep Space Network doppler noise is large (100 mm/sec), the onboard system makes a significant reduction in position error with respect to the planet for the Jupiter near-planet case. The reduction is from 33 km to 2.3 km. Second, during the interplanetary leg of the Jupiter mission, the onboard system can reduce ephemeris errors substantially from the 500-km level down to the vicinity of 150 km.

Outside of these two special cases, the onboard system can only be justified in relation to its potential use as a scientific instrument its interaction with the guidance process, and small improvements in the general body of navigation errors. Navigation results were not obtained for a Neptune mission, but from results obtained in the Phase A studies it can probably be concluded that the onboard system would make a stronger contribution to error reduction to a Neptune mission.

One of the more interesting results is the marked effect that highly accurate accelerometers have on the position and ephemeris errors, and on the spacecraft mass uncertainty. At the end of the Saturn mission interplanetary leg for example, the spacecraft position uncertainty is reduced by a three-accelerometer system to 82 km from an accelerometerless value of 1540 km. This occurs because two of the dominant error sources in the equations of motion are reduced by the accelerometers, namely the spacecraft mass uncertainty and the thrust vector misalignment. One accelerometer is much less useful than three mainly because it cannot distinguish thrust-vector misalignment from thrust-magnitude variations. However, the single accelerometer was assumed to be strapped down, and precision gimbaling of a single accelerometer with two-axis freedom and precision alignment with respect to the attitude control system would allow it to function in the same way as a set of three orthogonal strapped-down accelerometers.

The study results presented here have been parametric and aimed at establishing the limits of guidance errors, navigation uncertainties, and velocity corrections. A logical next step in preparing for outer planet or comet missions is to select specific missions and proceed with their optimization. From a guidance and navigation standpoint, this optimization would involve the design of an optimal guidance law and minimization of course correction fuel requirements. A specific navigation sensor configuration would be selected, constraints defined, and an optimal navigation measurement schedule established.

Recommendations

Further studies are indicated in the areas of guidance algorithm development, simulation structuring, and parametric variations. A number of guidance schemes should be investigated including optimal guidance. These would include allowing thrust to be switched on and off at times other than the nominal trajectory times as considered for the present scheme.

The statistical simulation should be restructured to produce the desired coupling between the guidance and navigation results. It should also be altered to eliminate the various numerical limitations which prevented the completion of a Neptune mission, caused several assorted cases to abort, and limited attitude control to a maximum of one arc minute. Part of the solution to the numerical problem involves the development of new covariance matrix propagation schemes.

In the area of onboard sensor studies, it would be useful to complete the parameter variations that were limited by numerical difficulties. In addition, the range of variation of some parameters should be extended. The Neptune results should be obtained.

The problem of accelerometer output sampling rate and period should be investigated to determine how problems of data processing of outputs onboard or on the Earth interact with error propagation from one set of measurements to the next, and how these factors affect the statistical modeling. The assumed white noise error model for accelerometers is proportioned to the sampling rate and the bias errors grow with time from last calibration.

Attitude control system importance should be determined by parameterising the limit on thrust-vector misalignment. Present results are representative of only a one-arc minute system.

In the onboard navigation area a number of results should be obtained. The effects of restricting the total navigational star field, and the types of navigation measurements should be investigated. Navigation errors are known to decrease with increased measurement frequency, but these effects have not been examined explicitly. Similarly the navigation measurement range from the target planet is known to have a strong effect on the value of the measurement, but the effects of restricting the range have not been determined. Curves showing error growth versus range and time would be helpful in this area.

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